(19) Europäisches Patentamt
European Patent Office
Office européen des brevets

(11) Publication number: **0 564 662 A1**

(12) **EUROPEAN PATENT APPLICATION published in accordance with Art. 158(3) EPC**

(21) Application number: **92922809.6**

(51) Int. Cl.^{5}: **B64C 21/08**

(22) Date of filing: **13.10.92**

(86) International application number: **PCT/RU92/00186**

(87) International publication number: **WO 93/08076 (29.04.93 93/11)**

(30) Priority: **14.10.91 SU 5004219**
14.10.91 SU 5004220

(43) Date of publication of application: **13.10.93 Bulletin 93/41**

(84) Designated Contracting States: **DE ES FR GB IT NL SE**

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(54) METHOD FOR CONTROLLING BOUNDARY LAYER ON AN AERODYNAMIC SURFACE OF A FLYING VEHICLE, AND A FLYING VEHICLE.

(57) A method for controlling the boundary layer by suction of the air from vortex chambers provided in the aft part of the aerodynamic surface of the flying vehicle (FV), wherein the speed of the air suction is regulated being first increased until the boundary layer joins the streamlined surface and then decreased until the pressure starts to drop at the aft part of the FV. An FV provided with a boundary layer control system comprising a number of vortex chambers (1) housing streamlined bodies (2) and connected by a common channel (9) through a receiver (8) to a low pressure source (4).

EP 0 564 662 A1

Technical Field

The present invention relates to aeronautics and has particular reference to methods for control of the boundary layer on the aerodynamic surface of an aircraft and to the construction of such an aircraft equipped with the boundary layer control (BLC) system.

Background Art

It is common knowledge that a measure of perfect aerodynamic performance characteristics of an aircraft depends on the aerodynamic fineness (lift-to-drag) ratio $K = C_L/C_D$, where C_L and C_D are the aerodynamic lift and the aerodynamic drag coefficients, respectively.

It ensues from the definition of the aerodynamic fineness ratio that in order to attain high values of K it is necessary either to reduce the aerodynamic drab of an aircraft or to increase its aerodynamic lift.

The lift can be increased by increasing the angle of attack of the aircraft lifting surfaces. However, with an increased angle of attack, a positive pressure gradient arises on the trailing surface of the wings directed along the airflow about the wing. At definite levels of said positive pressure gradient the airflow is incapable of moving against the positive pressure gradient due to its having inadequate kinetic energy near the wing surface and is separated from the latter.

Such an airflow separation results in a badly increased aerodynamic drug of the aircraft construction members streamed with the airflow and in a reduced lift of the aerodynamic lifting surfaces (i.e., wings and fuselages), that is, C_D increases, C_L decreases, with the result that the aerodynamic fineness ratio K is reduced too.

To ensure against aerodynamic stall and improve aircraft aerodynamic characteristics, as well as those of other aerial vehicles, the wall boundary layers of the airflow are sucked off, thus increasing the Kinetic energy of the wall boundary layer and its ability to overcome high pressure gradients.

The present state of the art knows a variety of practical solutions of the problem of the boundary layer control by sucking its off in the wall zone.

One of the prior-art methods for control of the boundary layer is known to effect air sucking off the wall zone through air-bleed orifices provided on the aerodynamic surface of an aircraft (cf. German Patent No. 1,273,338). The method is a highly energy-consuming one since air is bled from the wall zone in a direction normal to the boundary layer. The same disadvantage is inherent in a technical solution pertinent to an aircraft having its fuselage shaped as a thick short-span wing (cf. US Patent No, 3,037,321) eqipped with a boundary layer control device which appears as a boundary-layer-control manifold situated in the aft fuselage and communicating, through suction slots, with the wall airflow zone. Provision is made in the inlet portion of the manifold for a rarefying arrangement to establish rarefaction in the manifold with the aid of a bank of suction fans. The system is, however, too power-consuming due to high power input of the fan drives required for air suction off the low-pressure zones on the aircraft surface and blowing the air in the high-pressure zones near the trailing edge of the aircraft.

Moreover, the required fan input power is increased due to an excessively large amount of air sucked off the low-pressure zone. According to the laws governing boundary layer control, the amount of sucked-off or blow-in air required for establishing a nonseparated airflow increases intensely downstream of the airflow towards the trailing edge. In the boundary laer control discussed above the amount of sucked-off air is equal to the amount of the air blown in the vicinity of the trailing edge. According to the aforesaid law, the amount of sucked-off air in the control system under consideration is to be several times lower than that of the air blown-in near the aircraft rear. Any violation of the boundary layer control law results in higher power consumption for fan drive and affects adversely the aircraft aerodynamic efficiency. An excessively high suction results in a rise of the skin-friction drug.

More advanced are a method and devices for control of the boundary layer, wherein the wall air layer is sucked off using special chambers established in the trailing aerodynamic surface, vortex flows are created in the interior of said chambers, the direction of which in the wall portion of the chamber coinciding with the direction of the boundary layer whereby the velocity of the latter increases resulting in a nonseparated flow of an airfoil.

Known in the present state of the art is a device for the boundary layer control operating according to the method described before and appearing a number of vortex chambers located on the inner side of the airfoil and provided with holes arranged across the external airflow (cf. US Patent No. 4,671,474).

Vortex motion inside the chambers is maintained due to hydrodynamic interaction of the vortex motion inside the chamber with the external airflow in the zone of the suction holes and at the expense of the power of the air suction source.

However, said device suffers from some disadvantages the principal of which are sophisticated construction, high airfoil drab level, and highly power-consuming suction of the vortex flow.

High drag level results from a considerable airfoil drag due to poorly streamlined square shape of the chamber and on account of an increased skin-friction drab on the surface of vortex chambers.

Considerable power consumption for airflow suction is due to agreat resistance offered by communication lines connecting the vortex chambers to a low-pressure source. The throttling effect of the communication lines is especially high with respect to a sonic flow made use of in the known device. In addition, at low velocities of the external airflow and small values of the positive pressure gradient, the power system of the device operates in an uneconomic mode, this being due to the fact that the system is adjusted for the maximum airflow velocity and pressure gradient values and therefore sucks air in excess of the necessary amount, which leads to unjustified power consumption.

One more prior-art device for boundary layer control is known to have cylinder-shaped vortex chambers, whereby their profile (form) drag can be reduced (cf. British Patent No. 2,178,131). However, it is due to a small size of the suction slot communicating the airflow wall boundary layer with the vortex chamber that the area of interaction of the airflow in the vortex chamber with the external airflow has but inadequate extent to provide a necessary increase of the airflow velocity in the wall boundary layer thereof to prevent boundary layer separation in case of great positive pressure gradients.

Disclosure of the Invention

It is a principal object of the present invention to provide such a method for boundary layer control and such an aircraft construction equipped with a device for boundary layer control that ensure nonseparated air flow over an aerodynamic surface in most diverse flight modes of the aircraft with substantially low power consumption.

The foregoing object is accomplished due to the fact that the rate of bleed of the air sucked off the vortex chambers provided in the trailing edge of the aerodynamic surface is controlled as follows: first said rate is gradually increased until vortex flows appear in the chambers, said flows being attached to the boundary layer, then said rate is maintained at a level in which the attached airflow is preserved and a nonseparated flow about the aerodynamic surface occurs.

Used as a controlled parameter by which a nonseparated flow about the aerodynamic surface can be judged, may be the pressure in the aircraft rear, which proves to have a maximum value in case of a nonseparated flow. Thus, its decrease along with a reduction of the air bleed level is indicative that said reduction should be stopped.

Another peculiar feature of the proposed method resides in the fact that when the air is sucked off the vortex chambers at minimum rates, said suction is carried out by ejection bleeding, the air being ejected is consecutively added in a common passage to the air bled from the cells located in the direction towards the aircraft trailing-edge portion, that is, air ejection is effected in a direction from the last cell to the first one.

Formation of a common airflow in a direction from the rear cell to the first one enables one to make use of a pressure gradient set on the surface in the non-separated flow. The airflow sucked off the rear cell bleeds air from the other cells by virtue of the ejector effect, said cells being situated upstream of the airflow towards the first cell and having a lower pressure level than that effective in the rear cell.

A principal characteristic feature of an aircraft comprising a fuselage shaped as a lifting wing, a power plant in the form of turbojet engines, and a gas-dynamic boundary layer control system, incorporating a number of vortex chambers arranged consecutively in the wing trailing-edge portion and communicating with a low-pressure source, according to the present invention, consists in the provision of streamlined bodies located in the interior spaces of the vortex chambers and establishing annular ducts with the walls of the chambers, as well as in the provision of means for control of airflow velocity in said annular ducts in the boundary layer control system proposed herein.

Provision of said streamlined bodies together with equipping an aircraft with air suction rate control means makes it possible to facilitate establishing a stable attached vortex airflow and maintaining its circulation, as well as to increase the wall open portion of the vortex chambers, thereby extending the area of interaction between the airflow in the chambers and the boundary layer. Attempts to increase said area without the provision of streamlined bodies result in that the airflow in the chambers gets split into a number of flows, which affects very badly the efficiency of boundary layer control.

Another peculiar feature of the aircraft proposed herein is the communication of the vortex chambers with the low-pressure source through a common passage, wherein said means for control of airflow velocities in the chambers are located, said means appearing as ejectors and controlled butterfly dampers. Said ejectors appear as ducts communicating the vortex chambers with the flow-through portion of the common passage.

In a specific embodiment of the device said common passage may have a receiver in its inlet portion, said receiver being provided with a diffuser at its inlet. The front vortex chamber may be iso-

lated from the flow-through portion of the common passage and may communicate directly with the receiver through a duct admitting air from the receiver to the boundary layer towards the open portion of the interior of said chamber.

Provision of a receiver provides for normal functioning of the boundary layer control system with some of the aircraft engines shut down.

Still more peculiar feature of the invention is the provision of slots in the fuselage of the aircraft which communicate the flow-through portion of the common passage with the aircraft rarefaction area and accommodate controlled butterfly dampers. When the inlet portion of the common passage is shaped as a receiver said slots are located in the receiver top wall. Communication established between the common passage flow-through portion or the receiver and the aircraft rarefaction area enables one, under normal operating conditions of the system, to bleed part of the air being sucked off to the low-pressure area in the external airflow streaming the aerodynamic surface which cuts down power consumption for air suction. In addition, when the receiver communicates with the rarefied area this provides for partial functioning of the boundary layer control system with all the aircraft engines shut down.

Yet still more characteristic feature of the proposed aircraft, according to the invention, is the provision of a low-pressure source as an ejector located at the inlet or outlet of aircraft turbojet engine, or else at the outlet of said engine, or in the gas-flow passage thereof. This provides for an efficient source of air suction.

Brief Description of the Drawings

In what follows the present invention will now be illustrated by a detailed description of a specific exemplary embodiment thereof with reference to the accompanying drawings, wherein:

FIG. 1 is a longitudinal sectional view of an aircraft in the form of a thick aerodynamic airfoil provided with a boundary control device having four vortex chambers situated on the airfoil trailing-edge surface;

FIG. 2 is a sectional view of a vortex chamber with an ejecting duct, showing a velocity profile in the wall area applied in several airflow sections;

FIG. 3 is a sectional view of a vortex chamber first along the airflow, of a receiver, and of a part of the gas-dynamic passage communicating the vortex chamber with the low-pressure source; and

FIG. 4 illustrates pressure distribution over the surface of a thick aerodynamic airfoil in case of a separated flow over said airfoil (shown with a dotted line) and a nonseparated flow over said airfoil (shown with a solid line).

Best Method of Carrying Out the Invention

The device for boundary layer control consists of a number of vortex chambers 1 arranged in tandem in the aircraft rear. The interior spaces of the chambers accommodate streamlined bodies 2 which establish an annular duct 3 with the chamber walls. The chambers communicate with a low-pressure source 4, while each of the chambers is provided with an ejector appearing as a duct 5 communicating the chamber interior with the flow-through portion of a gas-dynamic passage common to all the ducts and communicating to the low-pressure soiurce 4. A first vortex chamber 6 may be isolated from said common passage (as shown in FIGS. 1 and 3), while the last chamber is devoid of ejector and its suction duct is in fact the initial portion of the gas-dynamic passage which is in effect a passage 7 and a receiver 8. The passage 7 merges with the receiver 8 through a diffuser 9. The interior of the receiver 8 communicates with the low-pressure area in the streaming-over airflow through slots 10 provided with controlled butterfly dampers 11. Controlled butterfly dampers 12, 13, 14 are provided in the gas-dynamic passage 7 and in the ejector ducts, respectively. An aircraft turbojet engine 15 with an ejector 16 may be used as a low-pressure source. The first vortex chamber 6 as along the airflow, when devoid of ejection air bleed, communicates with the receiver 8 through a duct 17.

The operating principle of the device for boundary layer control, according to the invention, is as follows.

Once the engine 15 has been started a low pressure is applied from the ejector 16 to the receiver 8, the diffuser 9, and the passage 7. The pressure level in the passage 7 increases towards the trailing-edge vortex chambers following approximately the same law as governs the pressure rise in the external airflow towards the trailing-edge aerodynamic surface.

The diffuser 9 communicating the passage 7 with the receiver 8 reduces the velocity of the air being sucked off and increases the pressure in the receiver 8, thereby improving the operating conditions of the ejector 16 at the inlet of the turbojet engine diffuser, thus decreasing the loss of the engine due to a reduced level of its throttling.

On putting the source of air bleed in a low pressure level extends to the interior spaces of the vortex chambers, whereby air flows over from the wall boundary area to the source of air bleed.

The gas velocity in the boundary layer increases with an increase level of air bleeding from

the interior spaces of the vortex chambers. As soon as the level of air bleeding reaches a certain value the boundary layer gets attached to the aircraft surface and a pressure with a positive gradient along the airfoil trailing edge is realized on that surface. The boundary layer attachment to the aircraft surface can be judged by the pressure measured in the airfoil trailing edge. An invariable pressure value on the airfoil surface when the air bleeding rate is increased is indicative of a non-separated airflow over said surface and of the onset of bound vortices in the vortex chambers. Checking for reliable boundary layer attachment to the aircraft surface against the value of the pressure on the airfoil trailing-edge surface is not, however, a single method. Used as such a control parameter may be the aircraft flying speed, inasmuch as boundary layer separation under steady flight conditions leads inescapably to reduction of the aircraft flying speed due to an increased aerodynamic drag.

Once the airflow has been attached the air bleeding rate is reduced, with the result that the air bleed intensity through the intake opening of the vortex chamber is reduced. Inasmuch as the bleed-lip leading edge A of the intake opening features a lower pressure than that on the trailing edge thereof, so as soon as the intensity of the air bleeding drops down to a certain level, air admission to the vortex chamber from the bleed-lip leading edge A ceases completely bit continues from the trailing edge B. Further reduction of the air bleed level leads to intensification of the air circulatory flow in the vortex chamber (that is, of the bound vortex), said flow being maintained by virtue of a pressure differential between the leading and trailing bleed-lip edges of the chamber intake opening. In this case the front portion of the chamber intake opening (along the edge A) functions as an air blow-in duct, while the rear portion of the intake opening (along the edge B) functions as an air suction-off duct.

Then the air suction-off level is reduced to minimum air bleed rate values at which non-separated airflow over the airfoil still takes place. Once the airflow has started separating the pressure level at the airfoil trailing-edge points (or the flying speed of the aircraft) starts dropping.

In order to reduce power consumption for the air bleed source an air ejecting suction from the vortex chambers is established. To this end, a common airflow is formed in the airfoil trailing edge by virtue of a positive pressure gradient realized on the airfoil surface when streamed without airflow separation, said common airflow being directed from the trailing edge cell to the first one. The pressure gradient adds to the airflow velocity, and the pressure at the outlet of the vortex chamber drops in a direction from the airfoil trailing edge. As a result, a pressure differential is built up at the inlet and outlet of the vortex chamber, required for gas ejection from the interior thereof.

Control of the process stated hereinbefore is effected with the air of the butterfly dampers 12, 13, and 14 and the ejectors 6.

With the aicraft taking off the butterfly dampers are opened completely so that air is vigorously sucked off (as shown with the dotted lines in FIG. 2) through the air bleed-lip edges A and B of the vortex chambers. In this case the amount of air sucked off in the boundary layer control system is too large and fails to provide an optimum operating mode of the system. The actual operating mode differs most widely from the optimum one at low flying speeds of the aicraft. However, such a mode facilitates, stable airflow attachment to the airfoil trailing-edge surface at large magnitudes of the angle of attack, gusts, lateral wind blows, and other perturbing factors. As the aircraft gains speed the difference between the actual mode of operation of the boundary layer control system from an optimum one gets narrower, whereas the angle of attack and other perturbing factors decrease, too.

The operating mode of the boundary layer control system gets optimized in a cruise mode of the aicraft. To this end, a search is carried out for an optimum position of the dampers 13 under the conditions of a maximum pressure on the airfoil trailing-edge surface or a maximum aircraft speed, the power rating of the engines remaining invariable and the other aircraft controls being in a fixed position. The fact that the prerequisite of a maximum aircraft flying speed is selected as a prescribed function makes it possible to take account of the influence of the air bleed level from the vortex chambers on pressure distribution over the aircraft aerodynamic surface, that is, to allow for the influence of the air bleed on the amount of the profile and induced drag. Furthermore, account is taken of the influence produced by the air bleed on the value of the friction force in the area of situation of the vortex cells and on the amount of thrust lost by the engines.

With the dampers 13 assuming an optimum position, stable bound vortices are established in the vortex chambers (indicated with the solid lines in FIG. 2), said vortices rotating under the action of a pressure differential effective in the external airflow attached to the air-foil surface. Inasmuch as the level of air bleed is in direct dependence on the position assumed by the damper 13, an optimum position of the latter corresponds to a minimized total drag, that is, to the flight conditions at the maximum aerodynamic fineness ratio.

At the final stage of aircraft landing the aerodynamic drag is to be increased, which can be

performed by partial airflow separation in the airfoil trailing-edge surface. To this aim, the level of air suction is decreased by closing the dampers 14 or the damper 12 in the passage 7. Opening of the slots 10 is also conducive to formation of a local flow separation on the airfoil trailing-edge surface.

In case of emergency shut-down of some of the aircraft engines, the running engines should ensure a required degree of rarefaction in the receiver. For this purpose the position of the dampers 13 in the gas passages of the running engines is to be changed, whereas the dampers 13 in the gas passages of the shut-down engines are to be closed.

In case of emergency shut-down of all the aircraft engines, all the dampers 13 should be closed and the dampers 11 should be opened. Under such conditions the vortex chambers continue operating under the action of a pressure differential between the area of maximum rarefaction on the aircraft fuselage and the pressure near the airfoil trailing edge. It is by virtue of said pressure differential that air flows along the diffuser duct 9 and thus air continues to be sucked off from the vortex cells with the aid of the matching ejector 5.

To provide normal operation of the turbojet engine 15 under starting conditions, use is made of the controlled dampers 11 situated in the slots 10 of the receiver 8. With the dampers 11 open rarefaction at the inlet of turbojet engine diffuser is reduced, thus preventing possible surge of the power plant compressor. Under nominal operating conditions of the boundary layer control system the controlled dampers 11 enable part of the sucked-off air to be bled from the receiver 8 through the slots 10 to the low-pressure area in the external airflow, thus cutting down power consumption for air suction.

Industrial Applicability

Design and experimental studies carried out on the basis of engineering and development work give evidence of a high level of technical and performance characteristics of an air-craft equipped, in particular, with the aforedescribed boundary layer control system, that is, the aerodynamic fineness ratio under cruise conditions of flight is from 18 to 25.

Claims

1. A method for control of the boundary layer on the aerodynamic surface of an aircraft by forming bound vortex flows in chambers established in the trailing-edge portion of the aerodynamic surface from which chambers air is sucked off, CHARACTERIZED in that the rate of air bleed from the chambers is controlled first by gradually increasing said rate until bound vortex flows appear in the cells, then by maintaining said rate at a level in which a nonseparated airflow over the aerodynamic surface is preserved.

2. A method according to Claim 1, CHARACTERIZED in that there is measured the pressure in the aircraft trailing-edge portion and a nonseparated airflow over the aerodynamic surface is judged by a maximum value of said pressure.

3. A method according to Claim 1, CHARACTERIZED in that at minimum rates of air suction, air is sucked by ejection bleeding from cells, the air being ejected is consecutively added in a common passage to the air bled from the cells located in the direction towards the trailing-edge portion of the aerodynamic surface.

4. An aircraft, comprising a fuselage shaped as a lifting wing, a power plant in the form of turbojet engines (15), and a gas-dynamic boundary layer control system, incorporating a number of vortex chambers (1) arranged consecutively in the wing trailing-edge portion, said chambers appearing as spaces open towards the boundary layer and communicating with a low-pressure source (4), CHARACTERIZED in that streamlined bodies (2) are accommodated in the interior spaces of the vortex chambers, an annular duct (3) being established between the walls of the vortex chamber and the surface of the streamlined body (2), while the boundary layer control system incorporates means for control of airflow velocity in said annular ducts.

5. An aircraft according to Claim 4, CHARACTERIZED in that the interior spaces of the vortex chambers (1) communicate with the low-pressure source (4) through a common passage (7), and the means for control of airflow velocity in the vortex chambers are shaped as ejectors (5) located in said common passage (7) and controlled butterfly dampers (12, 13) situated in said passage.

6. An aircraft according to Claim 5, CHARACTERIZED in that the ejectors appear as the ducts (5) communicating the interior spaces of the vortex chambers (1) with the flow-through portion of the common passage which communicates said chambers with the low-pressure

source.

7. An aircraft according to Claim 5, CHARACTERIZED in that the common passage communicating the interior spaces of the vortex chambers with the low-pressure source is provided in the inlet portion as a receiver (8) having a diffuser (9) at its inlet.

8. An aircraft according to Claim 5, CHARACTERIZED in that slots (10) with controlled butterfly dampers (11) are provided on the upper fuselage surface to communicate the flow-through portion of the common passage with the aircraft rarefaction area.

9. An aircraft according to Claims 7 and 8, CHARACTERIZED in that the slots (10) communicating the common passage with the rarefaction area are made in the top wall of the receiver (8).

10. An aircraft according to Claim 7, CHARACTERIZED in that at least one front vortex chamber (6) is isolated from the flow-through portion of the common passage and communicates with the receiver through a duct (17) which provides for tangential air admission from the receiver to the boundary layer towards the open portion of the interior of the vortex chamber.

11. An aircraft according to any of Claims 4 through 10, CHARACTERIZED in that the low-pressure source (4) is established by an ejector (16) at the inlet or outlet of the turbo-jet engine (15) of the aircraft, or in the gas passages thereof.

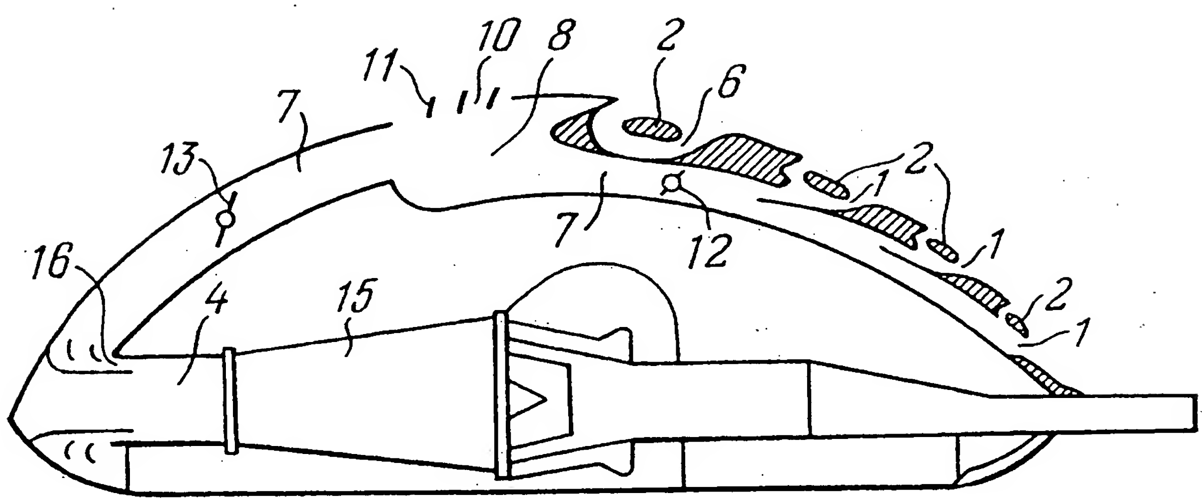

FIG. 1

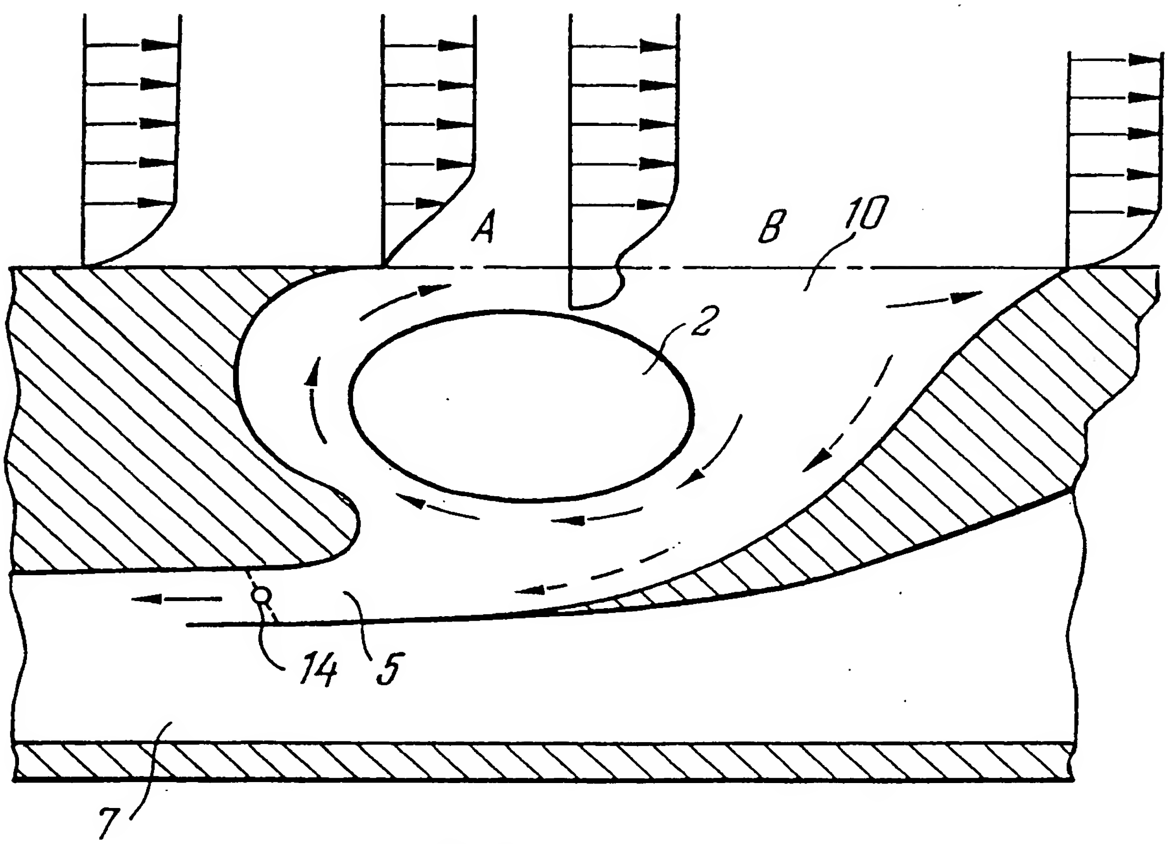

FIG. 2

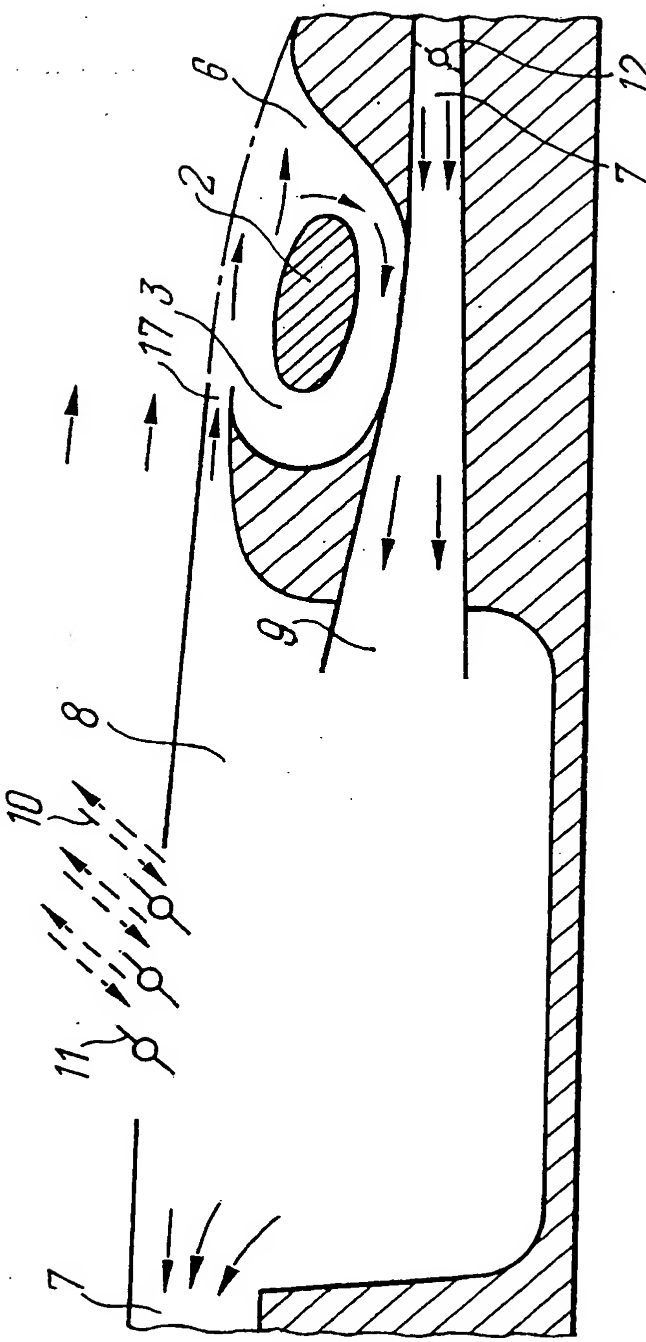

FIG. 3

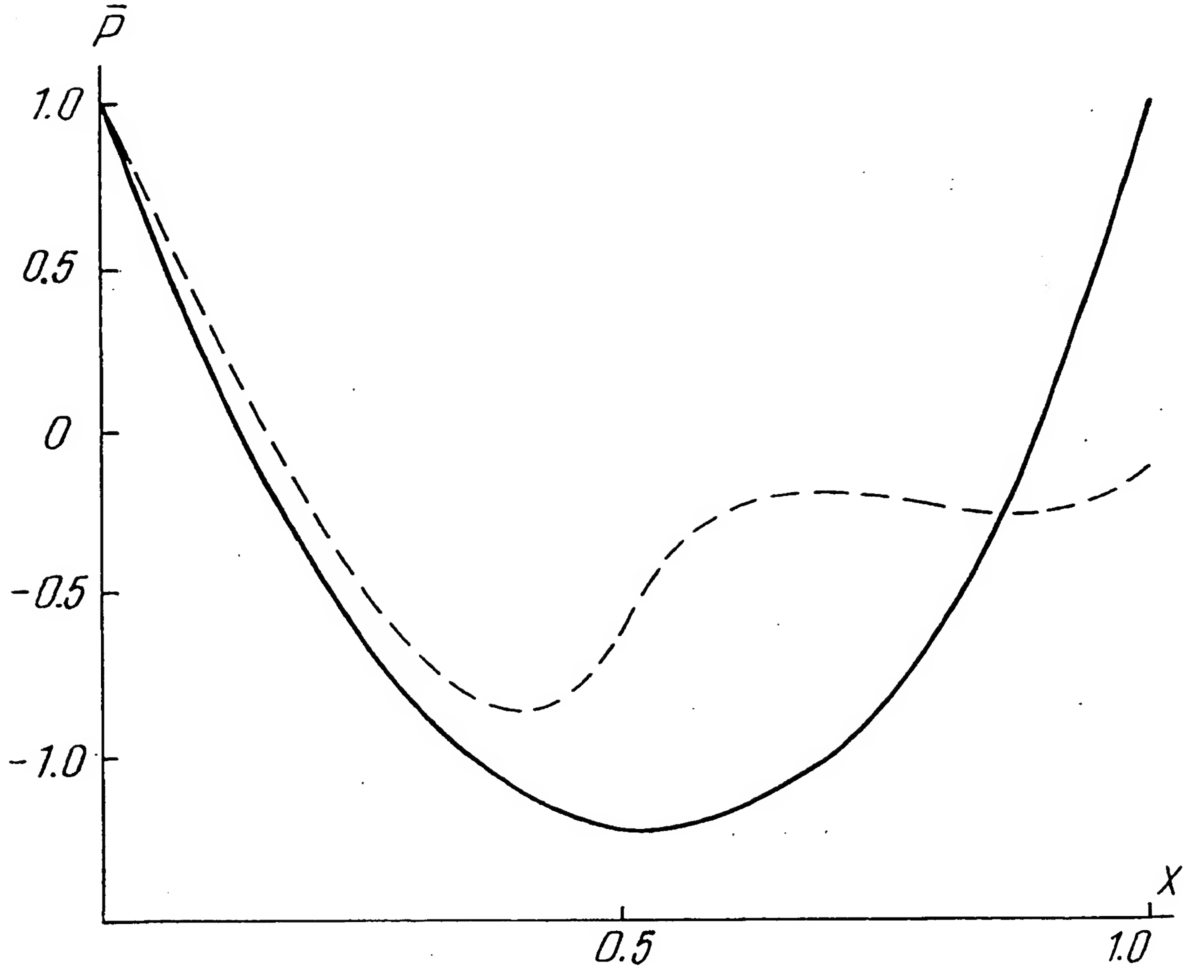

FIG.4

United States Patent [19]

Renshaw

[11] 3,790,107

[45] Feb. 5, 1974

[54] **BOUNDARY LAYER AIR CONTROL MECHANISM FOR AIRCRAFT**

[75] Inventor: **John H. Renshaw**, Marietta, Ga.

[73] Assignee: **Lockheed Aircraft Corporation**, Burbank, Calif.

[22] Filed: **Mar. 16, 1973**

[21] Appl. No.: **342,253**

[52] **U.S. Cl.**........................ **244/42 CC**, 244/42 DA

[51] **Int. Cl.**... **B64c 21/08**

[58] **Field of Search**..244/42 CC, 42 C, 42 D, 42 DA, 244/40 R, 41, 130

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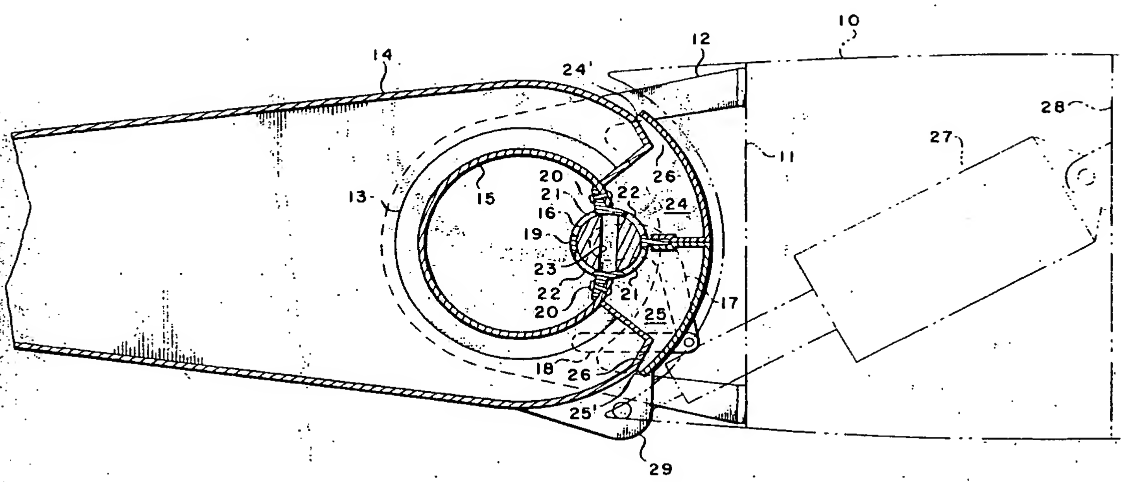
Primary Examiner—Duane A. Reger
Assistant Examiner—Barry L. Kelmachter
Attorney, Agent, or Firm—John J. Sullivan; Billy G. Corber

[57] **ABSTRACT**

Pressurized air is provided from an onboard aircraft source to a duct within an external control surface where it is selectively metered through a control valve to nozzles located on each side of the control surface. Movement of the control surface in one direction concurrently moves the control valve to pass the air exclusively to the nozzles on one side of the surface while movement of the control surface in the opposite direction concurrently moves the control valve to pass the air exclusively to the nozzles on the other side of the surface. Thus boundary layer air control is provided to the desired side of the control surface automatically from a single duct.

5 Claims, 4 Drawing Figures



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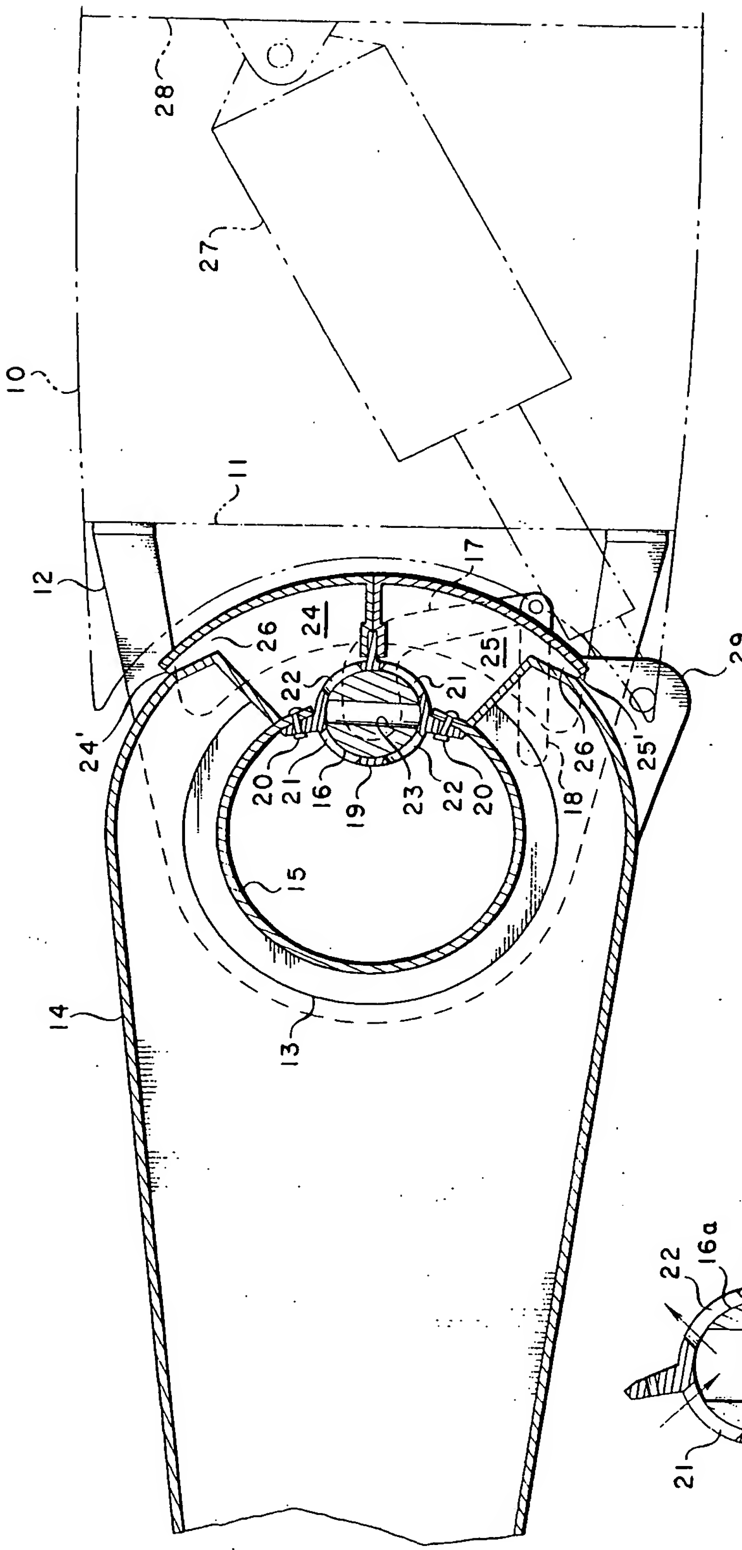

FIG. 1

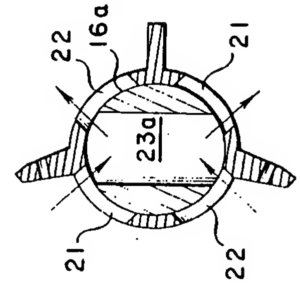

FIG. 4

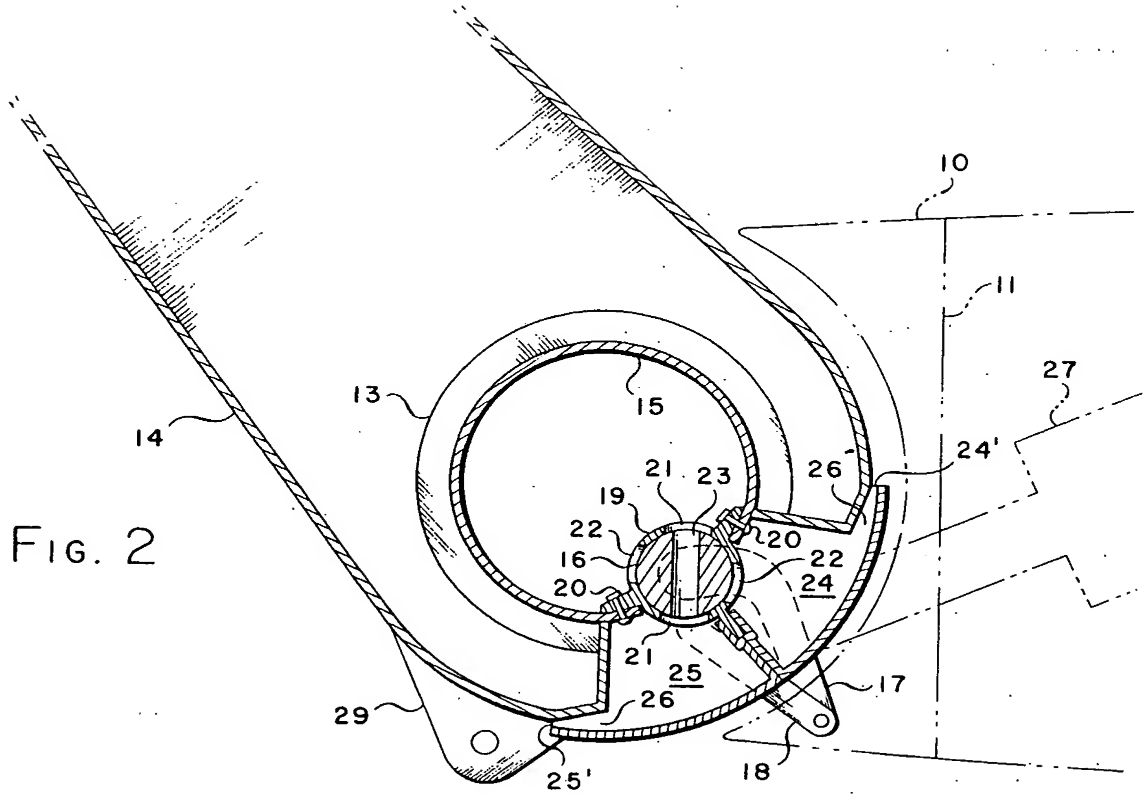
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FIG. 2

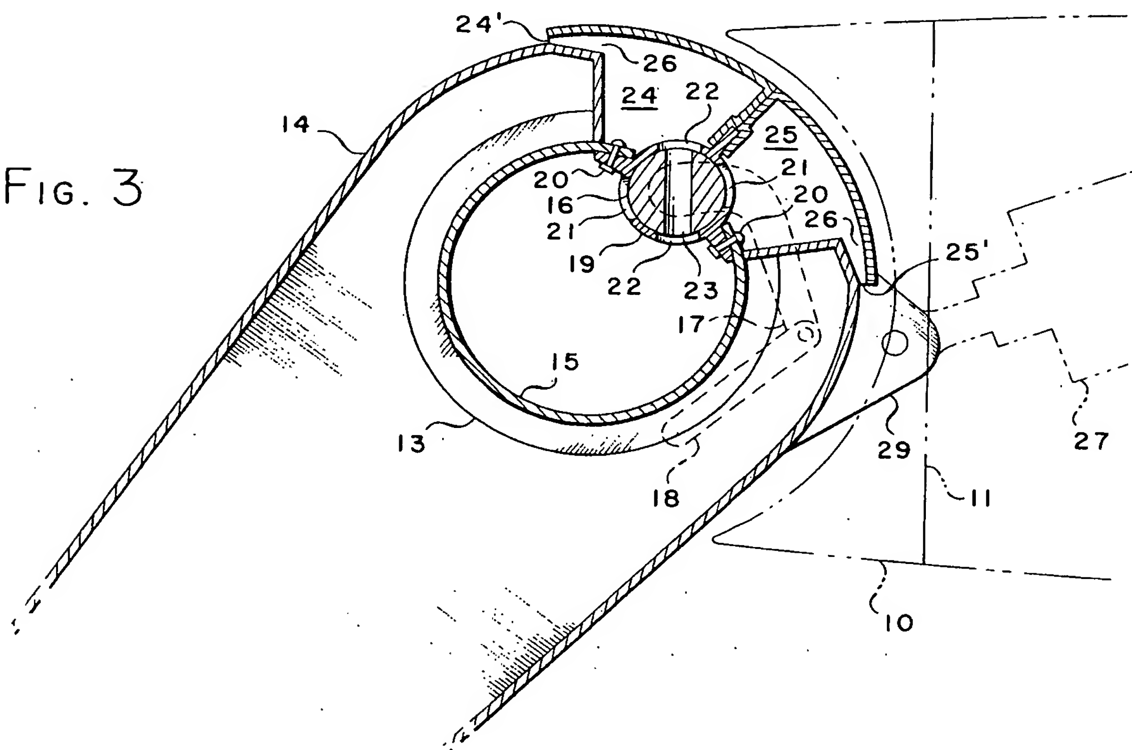
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FIG. 3

BOUNDARY LAYER AIR CONTROL MECHANISM FOR AIRCRAFT

This invention relates to boundary layer air control mechanisms as employed on aircraft to maintain airflow over the external surfaces thereof for the more efficient, aerodynamic functioning of the aircraft and more particularly to such a mechanism especially designed and constructed for use in conjunction with movable lift and/or control surfaces of aircraft whereby boundary layer air is selectively maintained on either side thereof.

While the present invention has general utility in boundary layer air control applications, it is particularly suited for aircraft having short take-off and landing (STOL) capabilities. In STOL aircraft comparatively large angular deflections of the external control surface member are required in order to produce lift and drag effects of the desired magnitude on the aircraft. Thus, boundary layer control has been generally accepted as necessary to assure the attachment of the airflow over the entire external surface of the control member when totally deflected. To this end, air jets or nozzles have been employed adjacent the hinge line of the control member to blow air obtained from a suitable onboard source in a generally parallel direction with respect to the external surface thereof. Since the most practical and convenient onboard source of air is the fan air of the engine, i.e., low pressure/low temperature air, relatively large diameter ducts are required to deliver the quantities of air needed to the nozzles. These ducts and related apparatus detract appreciably from the internal areas of the aircraft which might otherwise be available for fuel, cargo, and the like. This situation becomes aggravated where a requirement exists for boundary layer air control on both sides of the movable control surface.

In many cases, as for example in high performance aircraft where relatively thin wings are employed, the necessary ducting and/or multiple air ducts are either impossible or highly undesirable. Also, in these installations the valving to selectively discharge air over one or the other side of the control surface becomes complex and unreliable. At the same time, the drive mechanism for actuation of the control surface member must not be adversely affected or unduly burdened in its operation by duct and/or linkage obstructions.

The foregoing constitute the principal considerations underlying the present invention. Stated differently, this invention proposes to satisfy the requirements for boundary layer air control over both sides of a movable control surface of a relatively high performance aircraft by the provision of air nozzles adjacent the hinge line of the control surface operatively connected by ducting which accommodates low pressure/low temperature air. The proposed mechanism permits compact ducting and valving so as to require minimum space burden on the aircraft and a location minimizing its interference with the control surface movement and operation. Moreover, the valving and operating mechanism therefore is specifically designed to minimum moving parts resulting in a completely uncomplicated operation and maximum reliability.

More specifically, the boundary layer air control mechanism envisioned herein comprises a single, relatively large diameter duct common to nozzles associated with both sides of the control surface with a control valve to operatively connect a selected nozzle or nozzles on one side of said surface at a time to the duct. The actuator for the control valve is energized by the same agency employed to deflect the control surface so that its movement and direction is automatically coordinated with that of the control surface. The duct nozzles and associated control valve are all located internally of the control surface member and thereby preserve the interior of the associated aircraft component for fuel, cargo, payload use, as well as to remove the boundary layer air control elements from interference with drive actuators for the control surface member.

With the above and other objects in view as will be apparent, this invention consists in the construction, combination and arrangement of parts all as hereinafter more fully described, claimed, and illustrated in the accompanying drawings wherein:

FIG. 1 is a section taken through a portion of a relatively fixed component such as, for example, a wing, stabilizer, etc., of an aircraft and the adjacent relatively movable member, i.e., an aileron, flap, rudder, elevator, etc., to show a boundary layer air control mechanism designed and constructed in accordance with the teachings of this invention incorporated therein, the movable member being located in the neutral position;

FIG. 2 is a similar view with the movable member being located in one extreme position relative to the neutral position shown in FIG. 1;

FIG. 3 is a similar view with the movable member being located in the other extreme position relative to the neutral position shown in FIG. 1, and

FIG. 4 is a section similar to FIG. 1 of the control valve alone to show a modified form thereof.

Referring more particularly to the drawings, 10 designates fixed aircraft structure, for example, a fragment of the aft section of a fixed wing. The wing 10 is closed by a rear spar 11 and terminates aftwardly in a series of spaced extension ears 12 each mounting a hinge bearing 13 rotatably mounting an aileron 14 to form, in effect, an integral part thereof. To this end, each hinge bearing 13 is rotatable around a duct 15 secured to and carried internally by the aileron 14. The duct 15 is operatively connected in any conventional manner at one end to a source of air pressure, such as for example the engine fan (not shown) and is interrupted at selected points in its length to accommodate a valve 16. Each valve 16 is connected to the extension ear 12 by means of and through a horn 17 projecting integrally, in effect, from the valve 16 and a link 18 pivotally connected to the ear 12 and the outer end of the horn 17. Each valve 16 is thereby rotated bodily relative to the duct 15 when the aileron 14 is rotated on the hinge bearings 13 in a manner to be described.

The wall of the duct 15 is interrupted to accommodate each valve housing 19 which is secured as at 20 to the duct wall defining the interruption. The valve housing 19 is pierced by diametrically aligned openings 21 and 22 constituting ports for the escape of air from the duct 15. As shown in the embodiment of FIG. 1, these openings or ports 21 and 22 are normally closed by the valve 16, establishing the neutral position of the aileron 14. The valve 16 is pierced transversely by a passage 23 whereby rotation of the aileron 14 in one direction (FIG. 2) aligns the valve passage 23 with one pair of ports, for example, ports 21, while rotation of the aile-

ron 14 in the other direction (FIG. 3) aligns the valve passage 23 with the other pair of ports 22.

The aileron 14 adjacent the valve housing 19 is formed or otherwise provided with a pair of chambers 24 and 25. These chambers 24 and 25 are isolated one from the other with the chamber 24 communicating with the valve housing port 22 and the chamber 25 communicating with the valve housing port 21. Openings 26 in the wall of each chamber 24 and 25 direct air therefrom outwardly through an associated nozzle 24' and 25' respectively located tangentially of the opposite surface of the aileron 14.

In view of the foregoing structure and arrangement, downward rotation of the aileron 14 produces a counterclockwise rotation of the control valve housing 19 aligning the valve passage 23 with ports 22 permitting air to flow from the duct 15 into the upper chamber 24 of the aileron 14 for exit through the associated nozzle 24'. As the aileron 14 rotates, the nozzle 24' rotates with it reaching the optimum location when required, i.e., at maximum deflection.

Upon rotation of the aileron 14 in the other direction on the other hand the valve housing 19 is rotated in the opposite direction. The valve passage 23 is thereby aligned with ports 21 permitting air to flow from the duct 15 into the lower chamber 25 of the aileron 14 for ejection from the associated nozzle 25' in a similar manner to that described above.

It is, therefore, apparent that the movement of the aileron 14 to and from the extreme positions with respect to neutral automatically causes communication between the internal duct 15 and one or the other of the chambers 24 or 25 for discharge over the adjacent upper or lower surface of the aileron 14. Such movement of the aileron 14 is effected through a conventional actuator 27 mounted internally of the wing 10 secured, for example, at one end to wing structure, e.g., a wing intercostal member 28 and at its opposite end to a projecting ear 29 or its equivalent carried by the aileron 14. Thus as the actuator 27 is extended, the aileron 14 is deflected upward with respect to the neutral position and as it contracts, the aileron 14 moves down with respect to the neutral position. In either case, air from the duct 15 is caused to blow over the aileron surface that is opposite to the direction of such aileron movement.

Referring to FIG. 4, a slightly modified control valve 16*a* is shown. This valve 16a differs from the valve 16 previously described in that its transverse passage 23*a* is appreciably wider than the passage 23 of the FIG. 1 embodiment so as to normally overlap an equal portion of each of the ports 21 and 22. Thus, unlike FIG. 1, air in the duct 15 is free to flow symmetrically through ports 21 and 22 when the aileron 14 is located in the neutral position. Upon deflection of the aileron 14 as described, however, one of the ports 21 or 22 is gradually closed while the other port 21 or 22 is gradually opened. In the extreme position of deflection one port 21 or 22 is fully closed and the other port fully opened so that the entire air discharge is through only the selected port 21 or 22.

While the control mechanism herein contemplated has been illustrated and described in what is believed to be its best embodiment at the present time, the instant invention contemplates other embodiments under given conditions. Such other embodiments and variations are contemplated as fairly fall within the scope of the appended claims.

What is claimed is:

1. A boundary layer air control mechanism for aircraft having a movable external control surface comprising:
   an air duct mounted to fixed structure of the aircraft and disposed within said movable control surface;
   at least one pair of discrete chambers within said control surface each having at least one inlet port and at least one outlet port, each outlet port of one of said chamber pairs terminating in a nozzle adjacent one side of said control surface and each outlet port of the other of said chamber pairs terminating in a nozzle adjacent the other side of said control surface;
   a valve disposed between said duct and the inlet ports of each pair of chambers aforesaid operable to establish communication between said duct and each inlet port of either one of said chambers; and
   an actuator for said valve for the operation thereof whereby air in said duct is discharged through the nozzle on one side of said surface only.

2. The control mechanism of claim 1 wherein said fixed aircraft structure is a wing and said air duct is rotatably mounted in at least one bearing carried by said wing and extends spanwise along one end thereof.

3. The control mechanism of claim 1 wherein said valve includes a passage having a transverse dimension less than the distance between the associated inlet ports so that there is no communication between said duct and either one of said chambers except when the valve is operative as aforesaid.

4. The control mechanism of claim 1 wherein said valve includes a passage having a transverse dimension greater than the distance between the associated inlet ports so as to overlap an equal portion of each at all times except when the valve is operative as aforesaid.

5. The control mechanism of claim 1 wherein said actuator includes a first connection between said valve and said movable control surface for operation of the former concurrently with movement of the latter, and a second connection between said valve and said fixed structure for operation of the valve to discharge air through the nozzle on the side of said surface opposite to the direction of movement thereof.

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